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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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COMPARISON OF THE EXPERIMENTAL PRESSURE DISTRIBUTION ON AN NACA 0012 PROFILE AT HIGH SPEEDS WITH THAT CALCULATED BY THE RELAXATION METHOD

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SUMMARY

Pressure-distribution measurements were made on a 5-inch-chord NACA 0012 airfoil at zero angle of attack in the Langley rectangular high-speed tunnel, a 4- by 18-inch closed-throat tunnel, and compared with results calculated by Emmons for an equivalent airfoil-channel configuration by using the relaxation method. The comparison shows good agreement between theory and experiment at low Mach numbers. At higher Mach numbers, however, the theoretical calculations indicate higher negative pressure coefficients than were obtained experimentally. The spread between the values predicted by theory and experiment increases with increasing Mach number.

INTRODUCTION

The relaxation method has been proposed as a means of calculating pressure distributions on two-dimensional bodies at high speeds (reference 1). In this method of obtaining a numerical solution to the differential equations of motion of a nonviscous compressible fluid, the equations of motion are written in the finite-difference form and are applied to a network of points covering the flow field influenced by the body. The fineness of the network has a large effect on the accuracy to which the solution of the differential equations may be obtained.

The relaxation method has been used by Emmons to calculate the pressure distribution on an NACA 0012 profile at zero angle of attack in free air and in a channel having straight parallel sides 3.6 chords apart, at Mach numbers up to 0.75 (reference 2). In order to compare these calculations with a real flow, pressure-distribution tests of the NACA 0012 profile have been made in the Langley rectangular high-speed tunnel with a ratio of tunnel height to model chord of 3.6. The results of this experimental investigation are reported herein and are compared with the calculated pressure distributions obtained in reference 2 by the relaxation method.

APPARATUS AND TESTS

The tests were made in the Langley rectangular high-speed tunnel, a closed-throat, nonreturn, induction-type wind tunnel having a 4- by 18-inch test section (reference 3). The model tested was a 5-inch-chord NACA 0012 airfoil which spanned the 4-inch dimension of the test section. Pressures at 34 orifices on the model were measured by means of a multiple-tube manometer. Thirty of the orifices were distributed over the upper surface in order to provide a more accurate check on the theoretical results in the high Mach number range. The Reynolds numbers of the tests ranged from about 1.1×10^6 at Mach number 0.40 to 1.7×10^6 at Mach number 0.75.

Schlieren photographs of the flow over the model were also taken, a high-voltage spark of approximately 2-microseconds duration being used.

RESULTS

Figure 1 shows the experimental pressure distributions on the airfoil in the Langley rectangular high-speed tunnel compared with the pressure distributions calculated in reference 2 by the relaxation method for an equivalent airfoil-channel configuration. All the data are for the airfoil at zero angle of attack and at Mach numbers M from 0 to 0.75. The experimental curve at zero Mach number was obtained by extrapolating the result at Mach number 0.40 to Mach number 0 by means of the Prandtl-Glauert correction formula.

Figure 2 presents the theoretical and experimental curves of figure 1(d), together with two additional theoretical pressure distributions. One is the relaxation calculation of reference 2 for the airfoil in free air at Mach number 0.70. The other is an extrapolation of the relaxation calculation for the airfoil in free air at zero Mach number to Mach number 0.70 by the Von Kármán-Tsien compressibility correction. Both of these theoretical pressure distributions have been modified by the methods of reference 4 to correspond to constricted wind-tunnel conditions.

Schlieren photographs of the flow about the airfoil at zero angle of attack and at various Mach numbers between 0.70 and 0.78 are presented in figure 3.

DISCUSSION

A comparison of the curves of figure 1(a) shows that at zero Mach number there is excellent agreement between theory and experiment. At higher Mach numbers, however, the relaxation method gives higher negative pressure coefficients than were obtained experimentally.

Figure 1(d) indicates moderate differences at Mach number 0.70 between the experimental and the calculated pressure distributions. These differences may be due to the boundary layers that existed on the airfoil and on the wind-tunnel walls in the experimental tests, which the theory neglects, or to limitations of the relaxation method.

Figure 2 shows that a larger constriction correction than is indicated by the conventional methods of reference 4 would be necessary to bring the relaxation calculations for the airfoil-channel configuration and for the airfoil in free air into agreement; however, a correction smaller than that given by the conventional methods would be needed to make experiment agree with the relaxation calculation for free air. It is of interest to note that the pressure distribution calculated by applying the Von Kármán-Tsien compressibility correction to the relaxation solution for the airfoil at zero Mach number in free air and modifying for tunnel constriction approximates fairly well both the experimental pressure distribution and that calculated by the relaxation method for the airfoil at Mach number 0.70 in free air, modified for tunnel constriction. (See fig. 2.)

At the supercritical speeds the agreement between theory and experiment is poor, as shown by figures 1(e) and 1(f). At Mach number 0.75 the experimental peak negative pressure coefficient is considerably lower than that given by theory. The same factors as were mentioned in connection with Mach number 0.70 could have contributed to the disagreement; an additional factor is the assumption of a single shock in the relaxation calculation for the airfoil in the channel. It is easily conceivable that other solutions to the differential equations might have been obtained by introducing a series of weaker shocks into the flow field. The probability of the occurrence of a series of weaker shocks is indicated by the schlieren photographs of the flow at Mach numbers 0.73 and 0.75 (figs. 3(b) and 3(c)). This is the usual type of flow observed at Mach numbers not greatly in excess of the critical. A series of weaker shocks would produce a pressure recovery more in accordance with the experimental result.

The relaxation calculations for the airfoil in the channel at Mach number 0.75 predict an expansion immediately following the shock. (See fig. 1(f).) This phenomenon is to be expected in the perfect fluid

assumed in the relaxation method; as explained in reference 2, the expansion is necessary if the fluid is to follow the airfoil surface. However, this expansion is absent in the experimental pressure distribution, probably because of the occurrence of a series of weak shocks, together with the averaging effect of the airfoil boundary layer.

Langley Aeronautical Laboratory
National Advisory Committee for Aeronautics
Langley Air Force Base, Va., June 19, 1950

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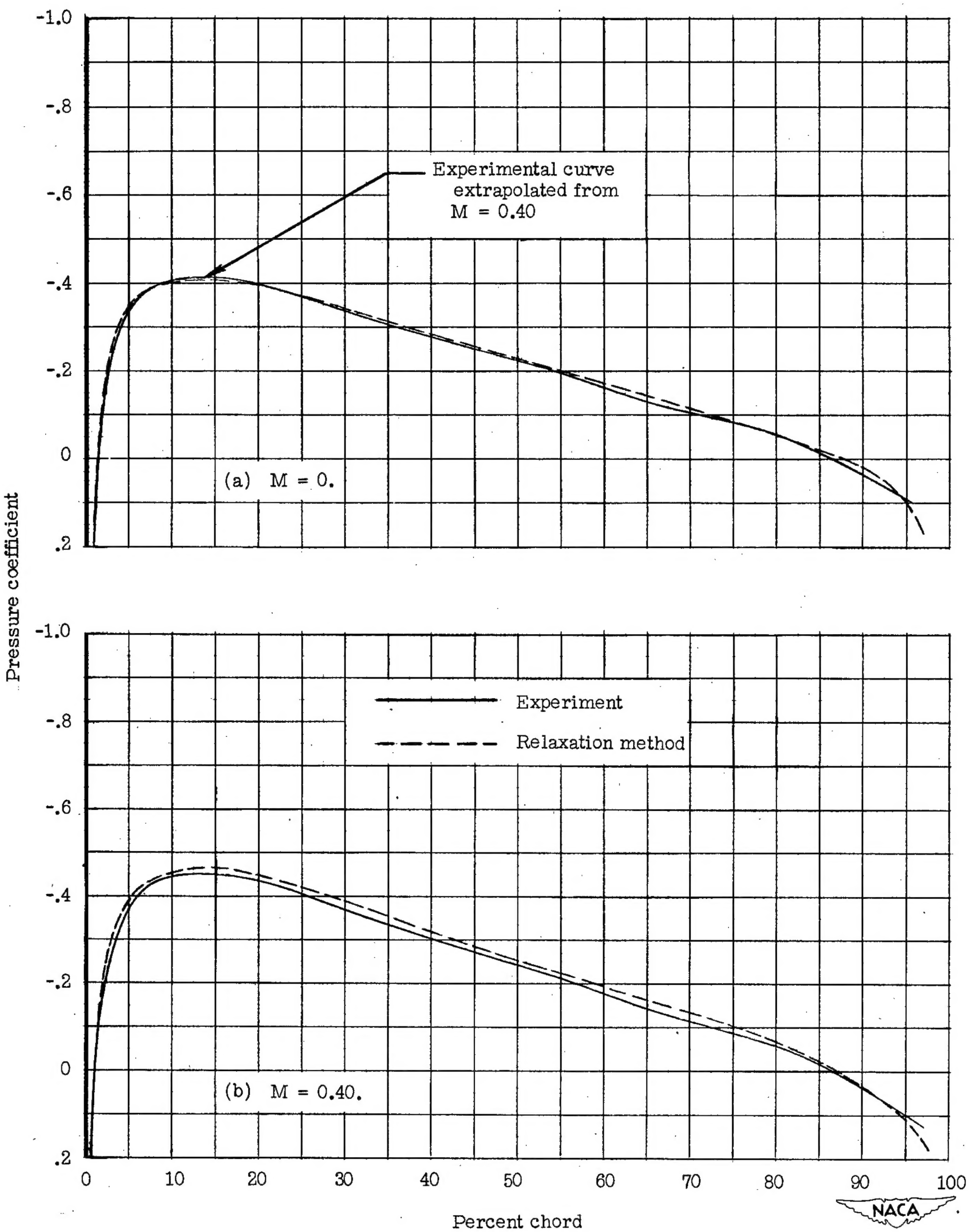


Figure 1.- Pressure distribution on an NACA 0012 airfoil in the Langley rectangular high-speed tunnel at zero angle of attack.

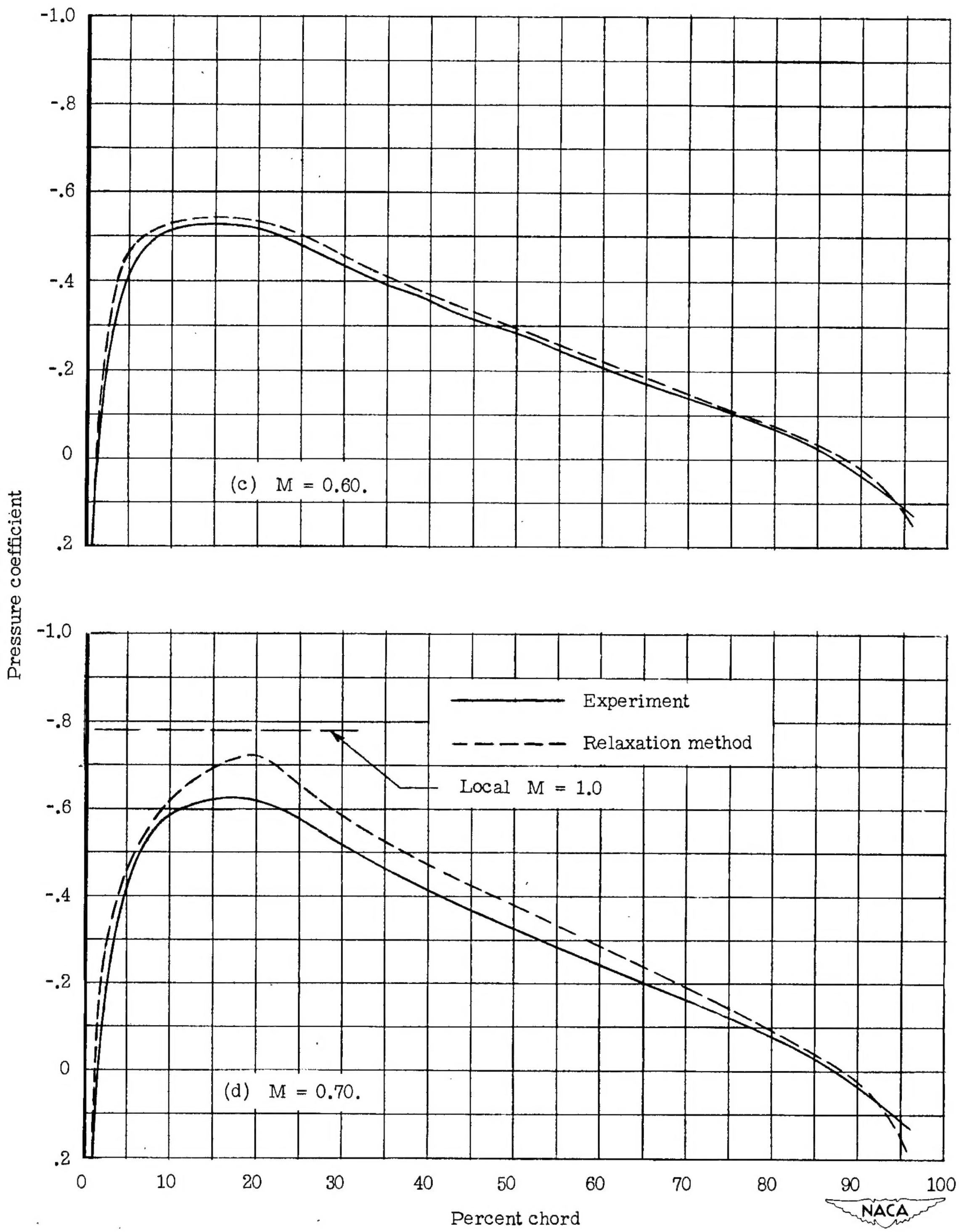


Figure 1.- Continued.

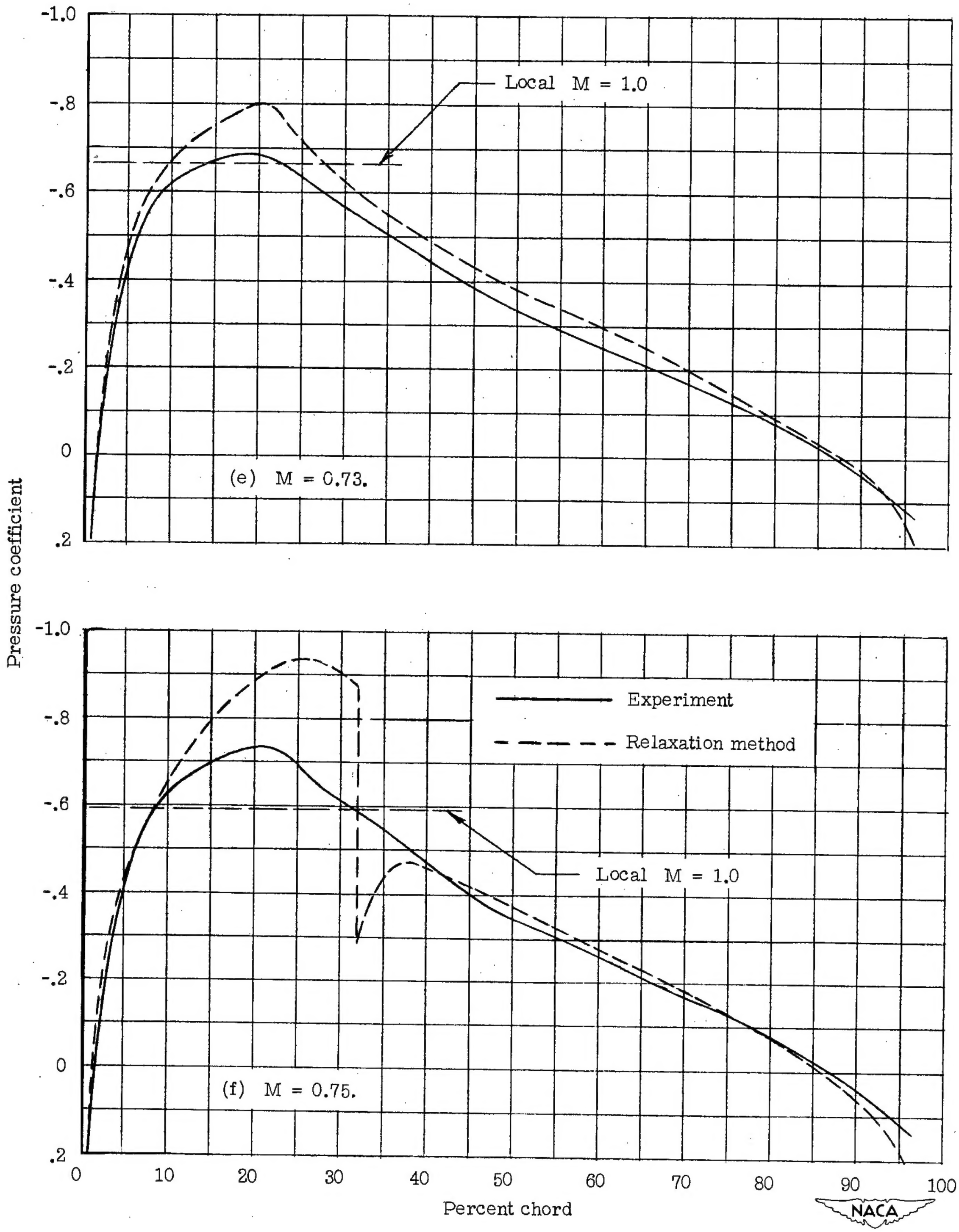


Figure 1.- Concluded.

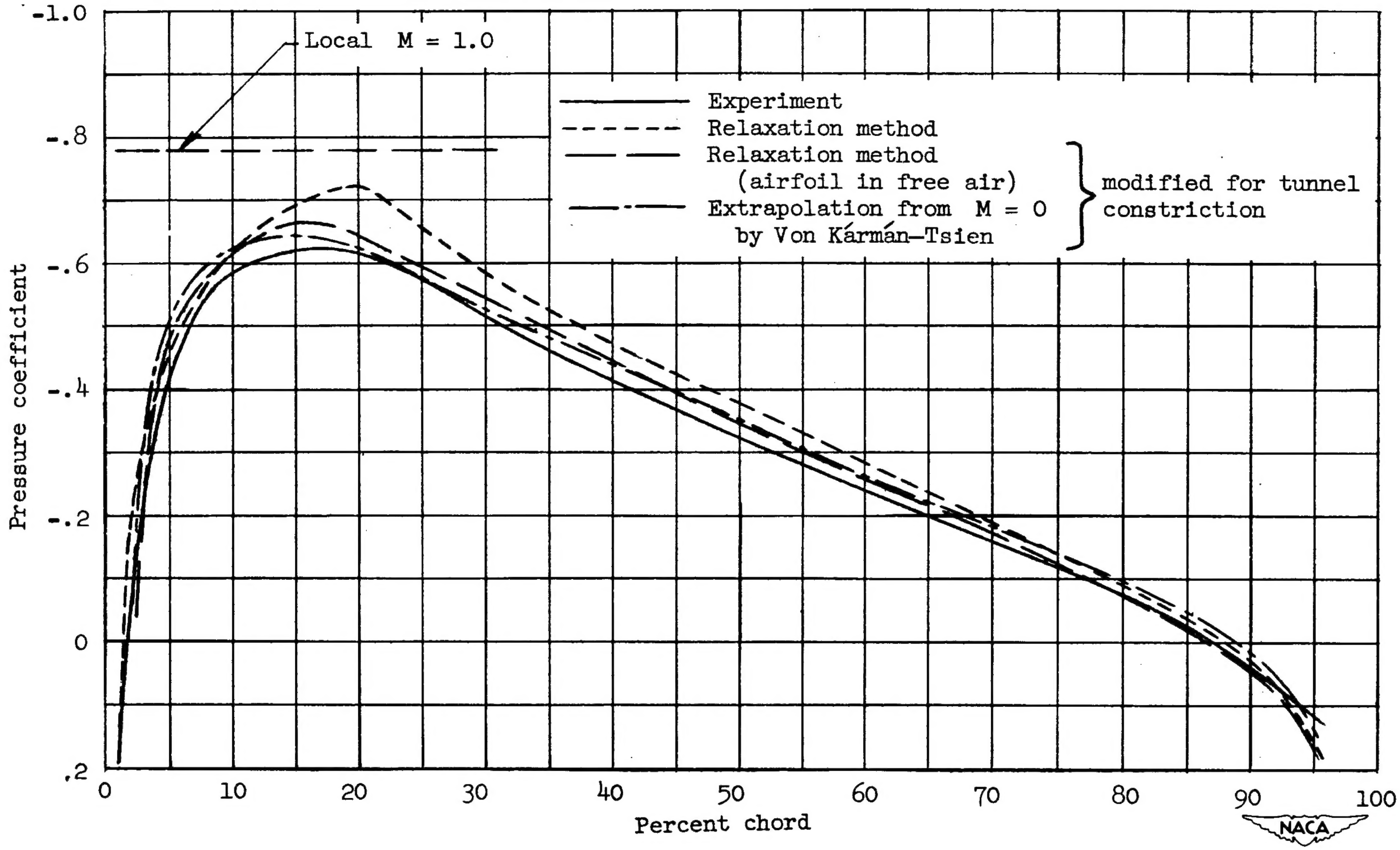
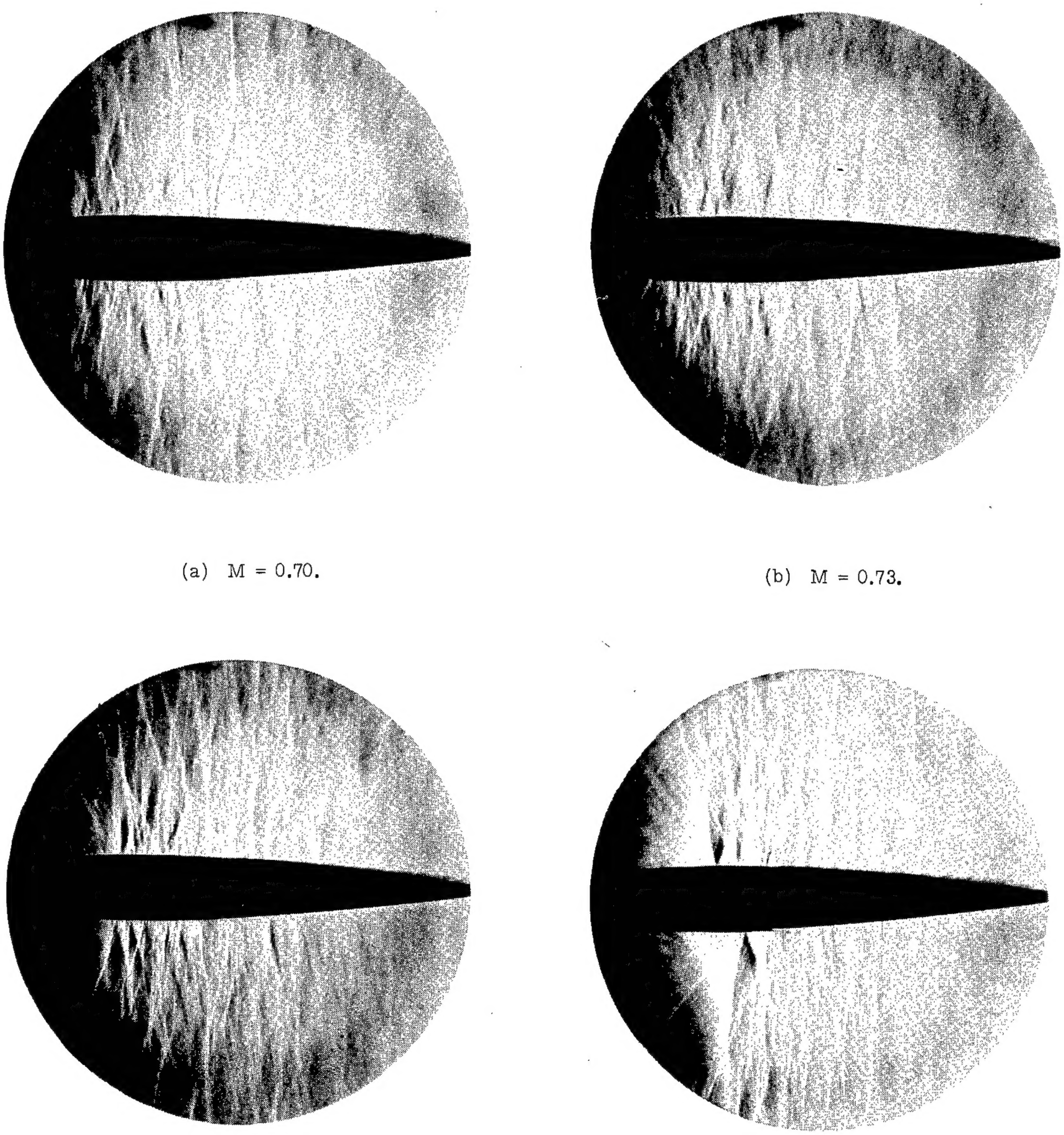


Figure 2.- Theoretical and experimental pressure distributions on an NACA 0012 profile at zero angle of attack in the Langley rectangular high-speed tunnel. M = 0.70.



(a) M = 0.70.

(b) M = 0.73.

(c) M = 0.75.

(d) M = 0.78.

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Figure 3.- Schlieren photographs of the flow past an NACA 0012 airfoil in the Langley rectangular high-speed tunnel at zero angle of attack.

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